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**A Titan Explorer Mission  
Utilizing Solar Electric Propulsion and  
Chemical Propulsion Systems**

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# **SOLAR ELECTRIC AND CHEMICAL PROPULSION FOR A TITAN MISSION**

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Mission and Systems analyses were performed for a Titan Explorer Mission scenario utilizing medium class launch vehicles, solar electric propulsion system (SEPS) for primary interplanetary propulsion, and chemical propulsion for capture at Titan. An examination of a range of system factors was performed to determine their affect on the payload delivery capability to Titan. The effect of varying the launch vehicle, solar array power, associated number of SEPS thrusters, chemical propellant combinations, tank liner thickness, and tank composite overwrap stress factor was investigated. This paper provides a parametric survey of the aforementioned set of system factors, delineating their affect on Titan payload delivery, as well as discussing aspects of planetary capture methodology.

**Introduction.** The space science community has significant interest in a planetary mission termed the Titan Explorer mission<sup>1</sup>. Due to this interest in a Titan mission, the In-Space propulsion program at Marshall Space Flight Center has supported an analysis program that was tasked to investigate the potential of performing this mission utilizing a medium launch vehicle including Delta and Atlas launch vehicles, a Solar Electric Propulsion System (SEPS) for interplanetary transfer, and a chemical propulsive capture at Titan. The mission entails several facets. First, a medium class launch vehicle places its payload on an optimal trajectory to reach Saturn. This trajectory includes a Venus Gravity Assist (VGA) in route to Saturn. A VGA, rather than an Earth Gravity Assist, was chosen for this analysis because of concerns of performing an Earth flyby with Plutonium based power sources (baseline Titan orbiter power source was chosen as radioisotope thermal generators). The launch vehicle payload includes a SEPS stage (separated from the remaining payload after completing the interplanetary propulsion), an orbiter that is placed in a circular 2000 km altitude orbit at Titan via chemical propulsion, and a lander (with aeroshell) that is released from the orbiter on a Titan direct entry trajectory well before the orbiter performs orbit insertion. The science payload consists of approximately 42 kg of instruments<sup>2</sup> on the orbiter and a 364 kg lander<sup>3</sup> that performs direct atmospheric entry at Titan. Thus, the total payload of lander and orbiter science instruments that must be delivered to Titan is approximately 406 kg. Note that only 42 kg of this payload is captured propulsively, with the lander separated prior to orbiter capture at Titan. Several propellant combinations were considered for this study. These combinations included state-of-art (SOA) Nitrogen Tetroxide (NTO) / Hydrazine (N<sub>2</sub>H<sub>4</sub>), Liquid Oxygen (LOX)/N<sub>2</sub>H<sub>4</sub>, Fluorine (F<sub>2</sub>)/N<sub>2</sub>H<sub>4</sub>, and a high Isp Monopropellant<sup>4</sup>. Advanced chemical propulsion tank technologies considered in this study included tank liner thickness and composite tank overwrap thickness.

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## Systems Analysis.

**Launch Vehicle Models.** Launch vehicle performance data<sup>5,6</sup> used to perform trajectory optimization are depicted in Figure 1. In this analysis, the launch vehicles investigated were the Delta-IV 4450, Atlas-V 421, and the Atlas-V 431. Data for the Delta-IV 4240 and the Delta-IV Heavy are provided for comparison purposes.

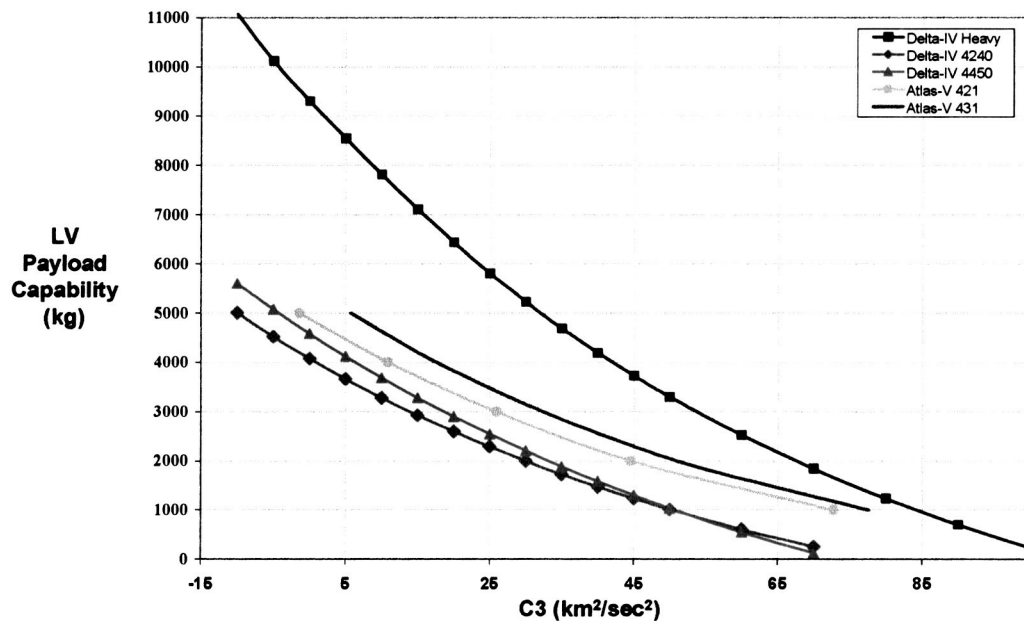


Figure 1 Launch Vehicle Performance Model

**SEPS System Models.** Use of high fidelity SEP vehicle synthesis models provided an estimate of the vehicle mass. Table 1 provides propulsion and power system definition. A graphic of the main SEPS system and subsystem elements modeled for this paper is shown in Figure 2. After computing the mass of the electric power (power generation, conditioning and distribution), propulsion (PPU, thrusters, gimbals, actuators), propellant (fluid management and tank thermal conditioning), and structures (bus, adaptors, mechanisms, thruster support, and tank support, component attachment), the remaining mass allocation represents the usable payload delivery capability to the Saturn destination. The relationship between the SEPS payload and other principle vehicle masses can be summarized as the difference between launch vehicle total payload to interplanetary injection and the wet mass of the SEPS stage. Determining the SEPS vehicle wet mass was one primary task of this study; the discussion that follows focuses on the primary power and propulsion systems.

Table 1 Baseline Propulsion and Power Systems Definition

System Definition	
Power	30 kWe at 1 AU EOL arrays; 25 kWe maximum into Ion Propulsion System; Multi-Junction GaAs arrays; Ultra-Flex design; housekeeping power is assumed to be covered by the 5 additional kWe array reserve.
Thrusters	4 thrusters with 1 spare; ~ 6 kWe @ 4000 sec Isp; NEXT design; Molybdenum grids.
PPU	4 PPUs with 1 spare; cross strapping PPUs; ~ 6.25 kWe maximum power, NEXT design; SOA heat pipe radiators.
Tank and Propellant	Tank fraction = 5%; supercritical Xe propellant
Propellant Management	NEXT design

**SEPS Power.** Large, high efficiency solar photovoltaic arrays provide propulsion power and vehicle housekeeping power (with the exception of battery power that must be provided for array deployment). An articulation of the arrays, in one axis relative to the sun, provides array feathering to control array temperature and to prevent the solar flux from exceeding a maximum allowable value on the arrays during the high solar intensity portion of the trajectory (e.g. spacecraft at  $< 1$  AU during Venus gravity assist). Prolonged solar array operation at high temperatures and array exposure to solar radiation degrades the efficiency of the photovoltaic cells; for this analysis a cell efficiency degradation factor of 2% average per year was applied. In addition, sizing the array area by 5% larger than required for the 30 kWe array output requirement provided further design margin. Able Engineering<sup>7</sup>, a solar array manufacturer, provided Ultra-Flex array modeling characteristics. The Ultra-Flex model represents the present state-of-the-art in lightweight solar array technology. Low Intensity Low Temperature (LILT<sup>8</sup>) effects were also accounted for in the array model.

**SEPS Propulsion.** A propulsion assumption for the SEPS vehicle includes an array of 5 NASA Evolutionary Xenon Thrusters (NEXT<sup>9,10</sup>) ion thrusters. SOA power processing units<sup>11</sup> (PPU) convert power from the solar array and deliver electrical power at proper voltage and current to the thruster array. The thruster elements consist of a set of thrusters, gimbals, actuators, a Digital Interface Control Unit (DICU), sun shield and support structure.

**Other SEP Systems.** Other SEPS vehicle subsystems play a critical role in the overall mass breakdown of the spacecraft, and hence affect delivered payload to the destination. Although this study accounted for these subsystems, no further examination is explicitly made herein. A diagram of the full SEP spacecraft system is depicted in Figure 2.

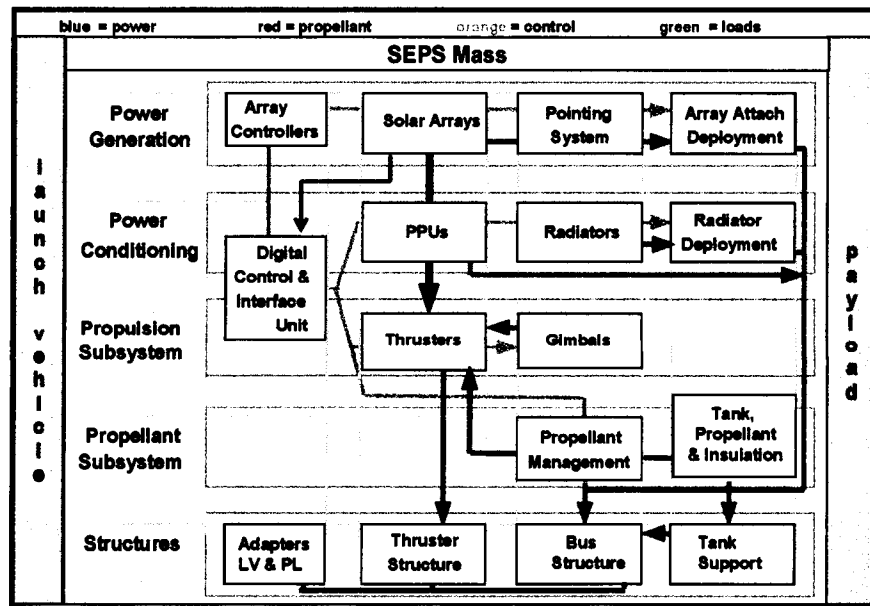


Figure 2 Total SEP System Definition Diagram

For each subsystem block indicated in Figure 2, the mass of the subsystem is computed, and then used to compute SEPS payload to Saturn.

SEPS systems analysis required various assumptions concerning mass and power margins, contingencies, and vehicle system redundancies. These margins, contingencies and redundancy assumptions are shown in Table 2.

**Table 2 Margins, Contingencies, Redundancies, and Other Assumptions**

<b>Contingencies</b>	
Launch Vehicle	2% nominal capacity (10% also investigated)
Propellant reserve, residual, navigation and trajectory corrections	10 % of deterministic propellant
Array End-of-Life Contingency	14% of baseline power mass
SEPS propulsion duty cycle	95%
<b>Margins</b>	
Dry Mass	30% of non-payload dry mass
Power	5% or baseline power mass
<b>Other</b>	
Redundancy	1 extra thruster, PPU, and DCIU
ACS during low thrust engine operation	Provided by Ion Propulsion System
ACS during engine-off	Provided by RCS

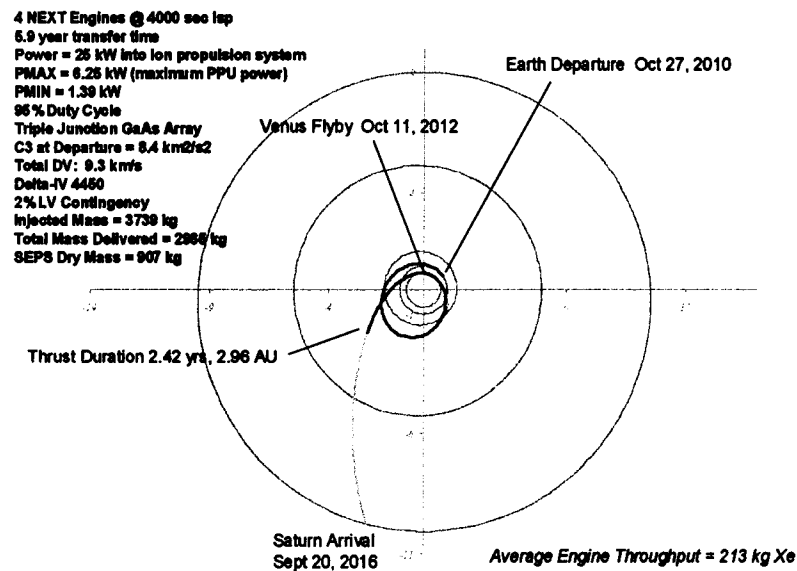
**Orbiter Chemical Propulsion System Models.** Planetary capture chemical propulsion system models consist of experience based data in the form of curve-fits of historical data and physics based models. For example, the composite overwrap tank model is scaled from the Advanced X-ray Astrophysics Facility (AXAF<sup>12</sup>) vehicle's composite tank. The baseline chemical capture stage consists of 2 main pressure-fed engines of nominally 100 psia chamber pressure. Main engines are sized for a thrust corresponding to an initial vehicle acceleration of 0.5 (about 200 lbf thrust). Other elements include the Thrust Vector Control (TVC) system, thermal conditioning, pressurization system, and the reaction control systems (RCS). Light weight tanks are operated with tank pressures on the order of 230 psia. Thermal conditioning is assumed to be supplied to tanks, lines, valves and thrusters. The pressurization system consists of high pressure, regulated gaseous He with redundant propellant management system controllers. The pressurant computation is based on the Van der Waals equation<sup>13</sup> in which the finite size of the Helium molecule must be considered in the model, due to the relatively high pressure assumption of 4500 psi. RCS consists of 16 thrusters, utilizing hydrazine monopropellant at 220 sec Isp. Other orbiter system models are not explicitly covered in this paper.

## **Mission Analysis**

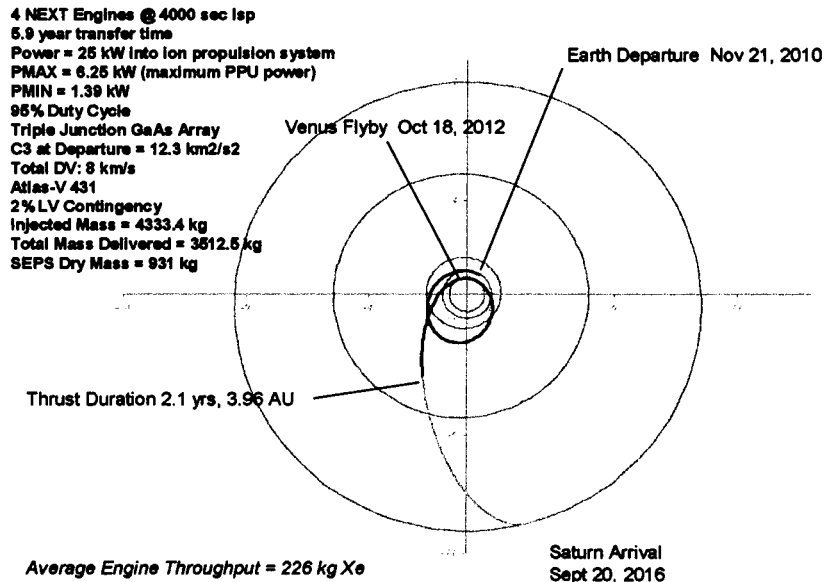
**Optimization of SEPS Trajectories.** For purposes of generating optimal trajectories to Saturn vicinity, the Solar Electric Propulsion Trajectory Optimization Program (SEPTOP<sup>14,15,16</sup>) was used. The trajectory optimization process includes launch vehicle throw weight capability (as a function of C3) as an optimization variable. SEPTOP was utilized to generate the interplanetary trajectories for a variety of relevant launch dates, trip times, departure C3's, arrival velocities, power levels and thruster combinations. Launch vehicle performance capabilities were previously provided in Figure 1. The SEPTOP trajectory optimization tool uses this previously mentioned launch vehicle performance data to determine the amount of delivered payload mass to the optimal C3 for the particular combination of transfer time, destination, power, and propulsion models. SEPS vehicle propellant load, transfer delta-velocity, thruster operation time, and thruster throttling and sequencing data are generated as well. Thruster models are imbedded into the SEPTOP computation process, allowing for global optimizations which factor in available array power, thruster efficiencies, and arrival energies among others. Constraints can be placed on major system elements such as maximum solar flux incident on the arrays and maximum/minimum thruster power levels, as examples.

**Earth-Venus-Saturn Trajectories.** The optimal gravity assist for the outer planet missions investigated tend to be of a class termed “energy pumping”. This term implies that the vehicle expends time in the inner solar system building energy before the Venus gravity assist occurs. Optimal energy gain occurs within the inner solar system as the vehicle is directed into a transfer path that takes it into solar distances of approximately 0.7 AU and 2 AU; within this high solar flux (inner solar system) region the vehicle receives most of its heliocentric velocity increase. This is referred to as an energy pumping maneuver. After the energy of the vehicle increases to the optimal value, the craft performs a Venus gravity assist that provides additional transfer energy to reach the destination within the prescribed transfer time. Typical energy pumping trajectories, shown in Figures 3 and 4, illustrate trajectories to the Titan destination. As can be seen from the diagrams, the spacecraft increases its potential energy by looping away from the sun, then upon reaching the optimal energy point, the vehicle moves toward the sun (increasing kinetic energy) and performs the Venus gravity assist. This maneuver provides a relatively large delta-v increase at Venus (on the order of 4.5 to 5.0 km/sec).

An Earth Gravity Assist (EGA) could have been chosen to perform the required kinetic energy gain. A larger gain in velocity could most likely have been realized with an EGA rather than with a VGA. Yet for this analysis, a VGA was chosen to eliminate concern of an Earth flyby with a plutonium based radioisotope power source on the spacecraft.



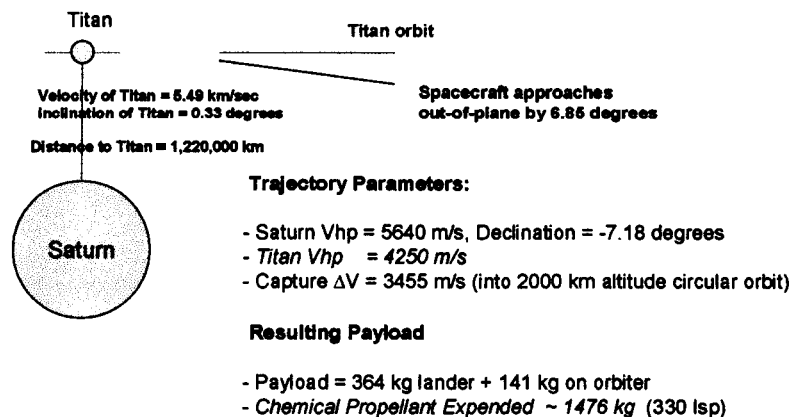
**Figure 3 Earth-Venus-Saturn Trajectory, 5.6 Year Transfer**



**Figure 4 Earth-Venus-Saturn Trajectory, 8.5 Year Transfer**

**Titan Capture Scenario.** Shown in Figure 5 is typical capture data for an Atlas-V 431 launch vehicle, 30 kWe SEPS interplanetary transfer stage, and NTO/N<sub>2</sub>H<sub>4</sub> propulsive capture at Titan. Aimpoint<sup>17</sup> approach calculations that accounted for Saturn interplanetary and Titan orbital geometry were performed and indicated a 3455 m/s delta-v to capture.

#### Atlas 431 LV, 8.5 Year Transfer



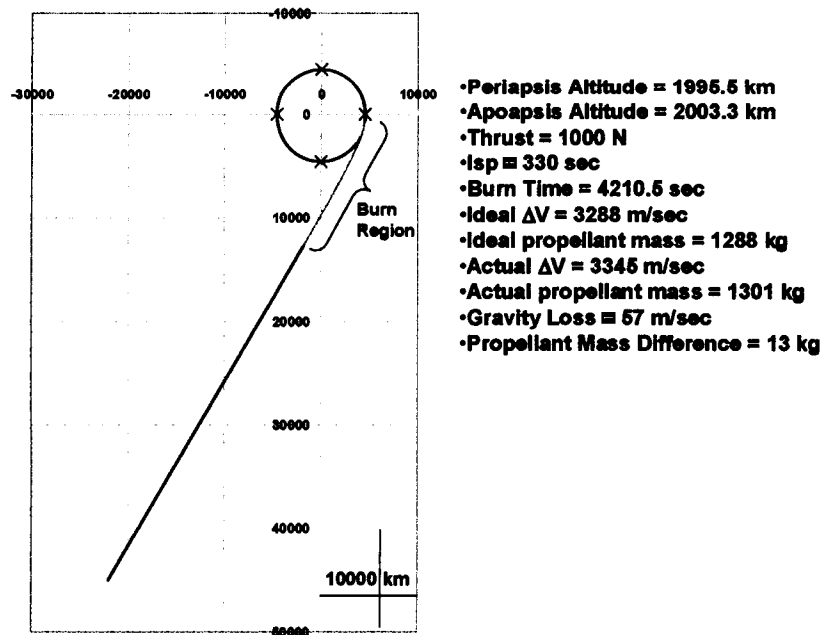
**Figure 5 Titan Capture Example**

A delta-v summary is shown in Table 3. This delta-v included g-loss assumptions (2.5% of ideal delta-v), navigation and trajectory correction error assumptions (20 m/sec), and contingencies for out-of-tangency Titan rendezvous (2% of ideal delta-v).

**Table 3 Typical NTO/N<sub>2</sub>H<sub>4</sub> Propulsion, Titan Capture Delta-V Budget**

<b>mu Titan</b>	9.027E+03	km <sup>3</sup> /s <sup>2</sup>
<b>radius Titan</b>	2575	km
<b>capture orbit altitude</b>	2000	km circular
<b>V<sub>hp</sub> relative to Titan</b>	4250	m/sec
<b>hyperbolic velocity at Titan periapsis</b>	4691	m/sec
<b>circular velocity at Titan capture orbit</b>	1405	m/sec
<b>Ideal capture delta velocity</b>	3287	m/sec
<b>contingency: orbit insertion uncertainty</b>	66	m/sec
<b>g-losses</b>	82	m/sec
<b>Nav &amp; trajectory correction delta-v</b>	20	m/sec
<b>Total delta-v</b>	3455	m/sec

**Gravity Loss Analysis for Titan Capture.** Analyses were performed to determine if adequate g-losses were accounted for in the baseline delta-V budget. To estimate the actual finite burn losses (g-losses), the Titan approach was modeled and an integrated trajectory was generated. As shown in Figures 6 and 7, the loss based on two approximately 100 lb thrust engines (330 sec Isp) was 57 m/sec (the difference between computed and ideal delta-v). The 57 m/sec loss turns out to be 1.7% greater than the ideal delta-v computed from the rocket equation. G-losses were fixed at 2.5% of ideal delta-v for the propulsion configuration assumed in this analysis. Thus, for SOA NTO/N<sub>2</sub>H<sub>4</sub> at 330 sec Isp, the g-losses were adequately accounted for in the delta-v budget.



**Figure 6 Trajectory Definition for Finite Burn Loss Analysis at Titan Capture**



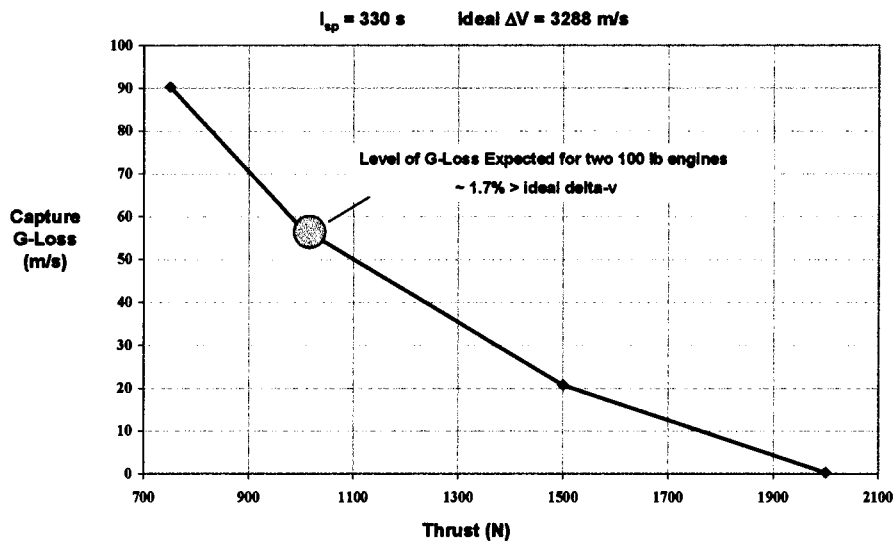


Figure 7 Finite Burn G-Loss Results, Titan Capture

## Analysis Results

**SEPS Payload Delivery to Saturn.** An examination of a range of launch vehicles determined the payload variation that could be expected for the mission evaluated. For the reference payloads targeted (and for significant payload variations about those reference payloads), the “Medium” class of launch vehicles provides adequate lift capability. A set of Boeing Delta-IV Medium and Lockheed Atlas-V Medium launch vehicles were selected for investigation. The launch vehicles examined included the Delta-IV Medium 4450, and the Atlas-V Medium 421 and 431. The resulting payloads to Saturn for a 4-engine 4000 sec Isp SEPS vehicle are depicted in Figure 8. Note that the average percent payload mass increase between consecutive cases averaged about 10% to 12%.

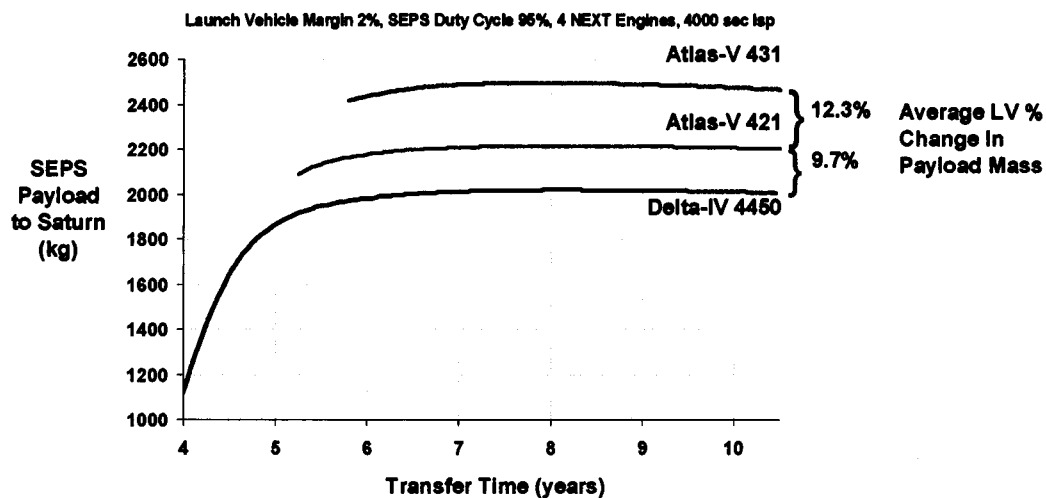


Figure 8 Variation in SEPS Payload to Saturn vs. Launch Vehicle and Transfer Time

For the power levels investigated, only small differences in Saturn payload performance was found for engine Isps of 3600 sec and 4000 sec. Yet as shown in Figure 9, a launch vehicle margin difference of 2% to 10% yielded an 8% increase in payload. As indicated further, a 10% margin provides adequate payload to the Titan destination for an Atlas-V 431 launch vehicle and SOA NTO/N<sub>2</sub>H<sub>4</sub>. It may be possible that if an EGA is assumed, the Atlas-V 421 may be able to place the 400 kg reference payload at Titan utilizing NTO/N<sub>2</sub>H<sub>4</sub> for Titan capture.

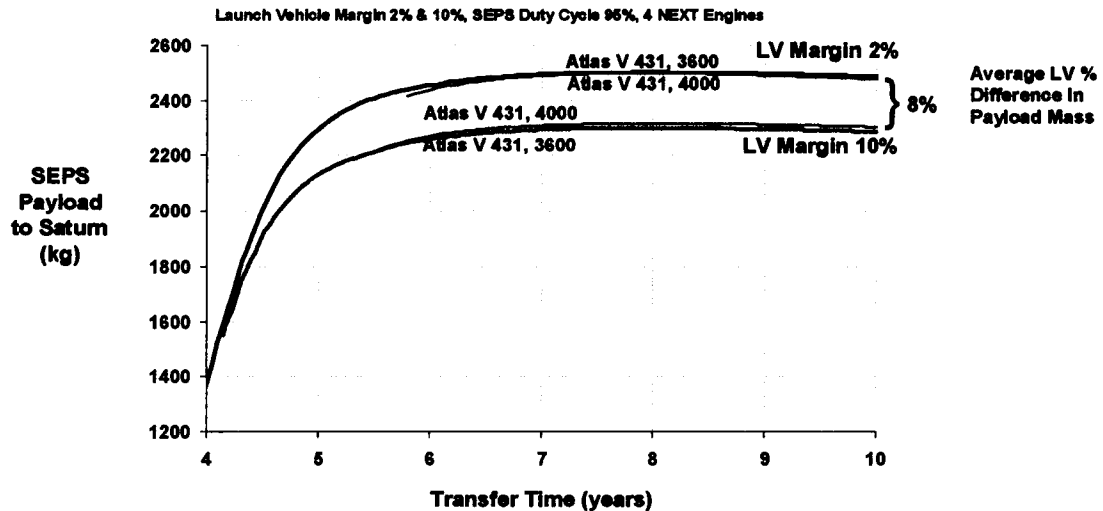


Figure 9 Impact of Launch Vehicle Contingency and Thruster Isp on Saturn Payload

Further consideration included a study to determine whether an increase in the number of SEPS thrusters (along with associated array power increase) could increase the payload to Saturn, see Figure 10. Results indicate the difference in the amount of SEPS payload delivered to Saturn for 4 and 5 thrusters is not large. As will be shown in upcoming results, little net Titan payload is realized for the additional complexity of 5 rather than 4 thrusters.

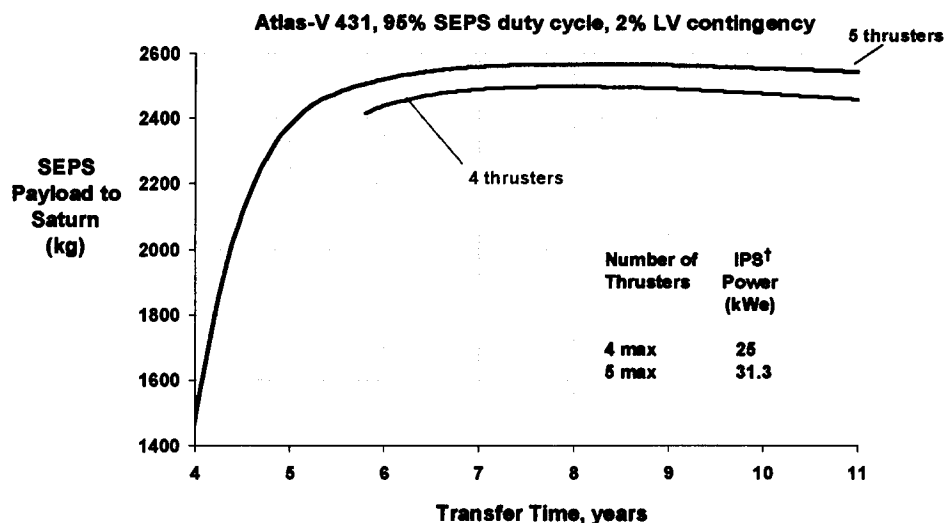


Figure 10 Impact of Varying Number of Thrusters and Associated Array Power on Saturn Payload

**Variation in Titan Payload for Range of Launch Vehicle and Chemical Propellant Combinations.** At Saturn distances the solar flux is too low to provide adequate power to the ion thrusters for propulsion; the chemical stage is therefore necessary to provide Titan capture propulsion. Capture analyses were done for a variety of final orbit altitudes; for the data presented in Figures 11-14, final Titan orbit was 2000 km altitude circular. Thrust levels were chosen such that gravity losses were kept to relatively low levels. The baseline chemical stage utilized storable NTO/N<sub>2</sub>H<sub>4</sub> propellants and off-the-shelf 100 lbf pressure fed engines.

Figures 11-14 illustrate payload delivered to Titan as a function of transfer time. Each figure is illustrative of a different propellant combination for the chemical propulsion capture stage. The reference Titan delivered payload for these missions is 406 kg, shown on Figures 11-14. This level is shown as the straight horizontal line on these figures and actual estimated payload delivery capability is shown as a series of curves; one each for three different launch vehicles. These delivered payload values vary with trip time and typically are maximum at an approximate transfer time of 8.5 years. As shown in Figure 11, a payload vs trip time plot indicates that the SEPS combined with the baseline NTO/N<sub>2</sub>H<sub>4</sub> chemical capture stage can deliver slightly more than 500 kg to Titan orbit, (100 kg more than the reference 406 kg payload) if an Atlas-V 431 is utilized. The minimum launch vehicle capability required for the 406 kg reference is just above the capability of the Delta-IV 4450. Figures 12 and 13 show corresponding data for O<sub>2</sub>/N<sub>2</sub>H<sub>4</sub> and F<sub>2</sub>/N<sub>2</sub>H<sub>4</sub> engine capture stages, respectively. Figure 14 contains corresponding data for a monopropellant engine chemical stage with an Isp of 275 sec.

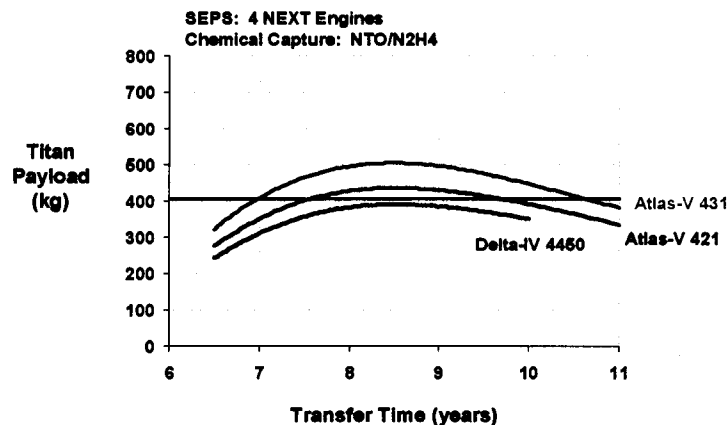


Figure 11 SOA Chemical Capture at Titan

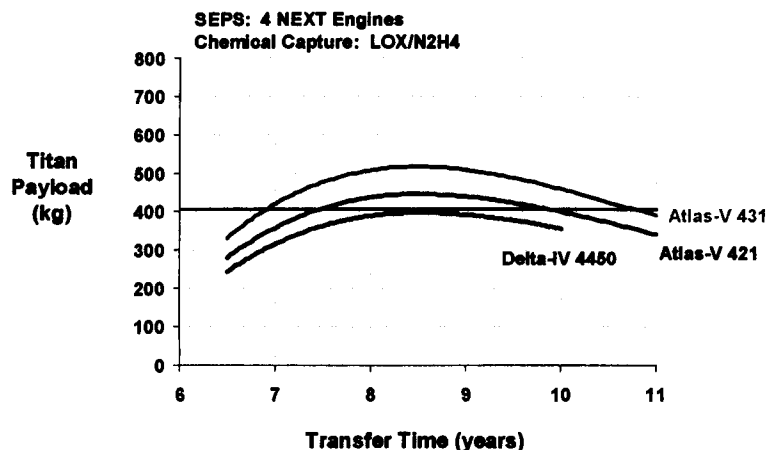


Figure 12 LOX/Hydrazine Capture at Titan

The highest payload capabilities can be achieved with the Fluorine based propellant<sup>18,19,20</sup>. Fluorine, however, has reactivity concerns in addition to toxicity issues. Fluorine engines were tested successfully by NASA and the Air Force during engine development programs of the 1960's, 1970's and early 1980's. A technology development program, however, would be necessary to bring this propulsion technology up to TRL level 9 status. The  $F_2/N_2H_4$  stage would provide an additional 100 kg payload increase over the baseline  $NTO/N_2H_4$  case with an Atlas-V 431 launch vehicle; payload to Titan in this case would be ~600kg. For the Atlas-V 421 and Delta-IV 4450, payloads are 520 and 460 kg respectively. As shown in Figure 14, a monopropellant operating at 275 sec Isp could not achieve the 406 kg reference payload. Should a 350 sec monopropellant version become available, payload would improve to about 600 kg, roughly equal to that of a Fluorine system. Monopropellant system performance, however, has not yet achieved Isp's in the mid 300 sec range. Current SOA maximum monopropellant Isp does not exceed the 280 sec range.

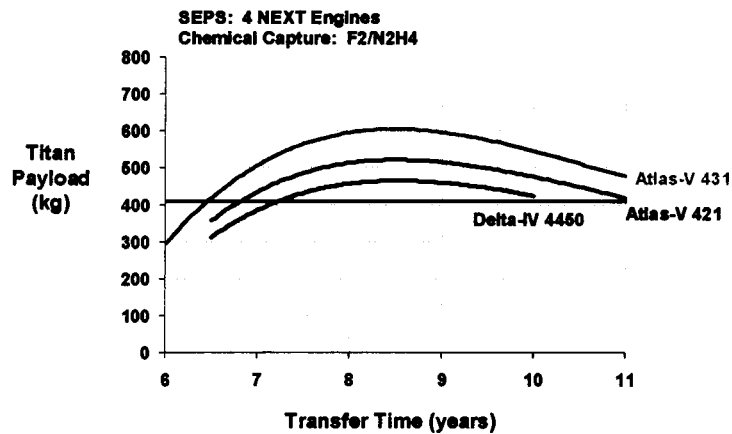


Figure 13 Fluorine/Hydrazine Capture at Titan

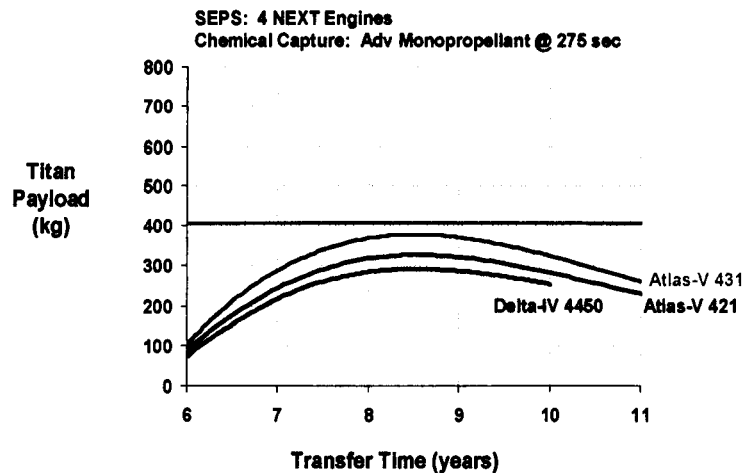


Figure 14 Advanced Monopropellant Titan Capture Results, Isp 275 sec

**Tank Technology Improvements Impact on Titan Payload.** Analyses were performed to ascertain the potential Titan payload increase that could be realized from tank technology improvements. A modest, yet significant increase in Titan payload of ~15 kg (over the 42 kg baseline) was found to be

possible by decreasing tank liner thickness by a factor of 6. Reducing composite tank overwrap weight by a factor of 30%, results in smaller % increases in payload.

**Analyses Conclusions.** Provided is a summary of the conclusions that were derived from the analyses performed in this study:

For the baseline storable propellant NTO/N<sub>2</sub>H<sub>4</sub> propulsion, the following conclusions can be made: The Atlas-V 421 class LV is the minimum launch vehicle required to deliver the DRM payload to Titan; the Atlas-V 431 can deliver the DRM payload for LV margin greater than 10%; Optimal payload delivery was found for 5 SEPS engines, but 4 SEPS engines may provide sufficient margin considering additional complexity; The off-the-shelf propulsion system provides advantages of many years of development and operation heritage.

For Advanced Fluorine F<sub>2</sub>/N<sub>2</sub>H<sub>4</sub> propulsion: System provides significant performance increase over NTO/N<sub>2</sub>H<sub>4</sub>; System would require the expense of Fluorine engine technology development; Fluorine engines tested successfully in 1960 and 1970's programs

For LOX/N<sub>2</sub>H<sub>4</sub> Propulsion: Provides some performance improvement over NTO/N<sub>2</sub>H<sub>4</sub>, but improvement is marginal and probably does not warrant a new engine development program for pressure fed engines.

Monopropellant based propulsion systems provide no improvements over SOA chemical unless the Isp is significantly greater than 275 sec.

Tank technology improvements can provide minimal improvements in performance by significantly decreasing tank liner thickness and reducing the pressure factor. Manufacturing and handling requirements, however, may dictate minimum tank and liner thicknesses; 7 to 10 year mission durations may impact minimum thickness allowances as well.

This analysis indicates that SEP/Chemical systems can deliver the 406 kg payload reference payload to Titan using "conventional" chemical technology, with the following allowances: With SEPS and SOA propulsion capture at Titan, transfer times of approximately 8 to 8.5 years will be required and a launch vehicle of performance in the Atlas-V 421 to 431 range will be necessary.

## **Acknowledgements**

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